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Subject: Bi-Monthly Progress Report for the period 1 September thru 1 November 1969. The work accomplished during this period will be reported under the following headings:

Task 1. Performance Investigation

Task 2. Heat Transfer Investigation

Task 3. Combustion Stability Investigation

I. INTRODUCTION

During this report period research in the following areas has been conducted. The details of the research are presented in the following sections.

A. One experimental firing was conducted to evaluate the modified pulse gun. The noise level on the analog tape machine reached unacceptable levels and it was necessary to have it reworked by Honeywell-Denver to correct this condition.

B. Research on heat transfer has been concentrated in a literature review to search for additional methods of measurement of heat flux under transient conditions.

C. In an effort to obtain a better insight into the ablative throat area variation effect on performance, a copper nozzle was fabricated and coated with ZrO_2, as previously reported. One experimental firing was made with this nozzle during this report period.

Effort is continuing on the improvement of the quality of the reduced data for experimental Runs 7 through 24. In addition to the removal of Runs 7 and 11 mentioned in the previous progress report, Runs 8, 9 and 15 have been deleted from the analysis. Because these were short runs, steady state conditions were not achieved.

II. Status of Work in Progress

A. Task 1. Performance Investigation

An experimental firing was conducted using the ZrO_2 coated copper nozzle. The post fire examination of the nozzle showed severe erosion at the throat (see Fig. 1 for photo of sectioned nozzle) indicating a failure of the ZrO_2 coating. A cross section sample was cut from the convergent section and micrographs were made as shown in Fig. 2. Examination of the photomicrographs revealed a distinct boundary between the ZrO_2 coating and the copper which indicates little of no diffusion bonding. The spheroidal nature of the ZrO_2 noted in the photos also indicates poor bonding within the coating itself. The loss of the ZrO_2 coating from the nozzle surface exposed the unprotected copper to the combustion gases which previous analysis has shown would not withstand the heat flux levels experienced in this engine for a sufficient length of time to acquire data.

The design of the single element quadlet injector is being formulated. The face of the injector is to be transpiration cooled by oxidizer flow. At the present time, the transpiration material is being selected to yield the desired flow rate, pressure drop, and structural integrity required for this application. In order to correctly match the flow rate and pressure drop, a dimensionless correlation is being developed for Rigimesh which will allow us to use air data to predict flow parameters of N_2O_4. The following dimensionless parameters were established:

$$\pi_1 = \frac{e\Delta p}{\dot{m}^2}$$

$$\pi_2 = \frac{n}{t\dot{m}}$$

where

e = density

n = viscosity

t = thickness

$\dot{m}$ = mass flow rate per area per second

Δp = pressure drop

Preliminary calculations (made by using air and water data received from Porious Media Inc.) indicate that these parameters will graph as a straight line on log-log paper.

The basic design of the injector has been established with the selection of the tube sizes and Rigimesh opening size remaining. Rigimesh is the favored media as there are some samples of this material available at the Jet Propulsion Center for immediate use. If the correlation proves successful, additional transpiration cooling data may be obtained by changing the Rigimesh injector face. The face cooled injector will also allow runs of longer duration which may be necessary for the acquisition of more reliable heat transfer data.

For previous reported performance calculations the nozzle throat area was assumed to ablate linearly in the interval of time when the chamber pressure exceeded the 3000 psia (this pressure is achieved approximately 0.02 seconds after startup). Other studies, such as those of Peterson NASA, indicate that throat area first contracts during the early part of the run before final ablative expansion to the post-fire area. A study is presently being conducted employing the experimental data to obtain a more precise relationship for the throat area as a function of time.

Significant differences were discovered between the thrust coefficients for the short runs and those of the longer runs. The thrust coefficient for the shorter experimental runs were shown to be significantly lower than the expected value of 1.65, while agreement was much better for the longer runs. A possible explanation might be that the averaging characteristics of the thrust measuring load cell and recorder combination is not the same for the shorter run durations as it is for the longer duration runs. This subject is currently under study and present indications are that a reproducible trend is indicated and that a consistant correction can be applied to the shorter duration runs.

In this report period two new computer programs were written to take advantage of the CALCOMP plotter at the Computer Science Center. The first program will plot the reduced run data in graphic form. The channels plotted include the chamber pressure, thrust, fuel and oxidizer flows and photocon response. The other plot routine is incorporated with the C* and Isp values to show the variations which occurred. Total flow and mixture ratio are also plotted. Analysis of the test runs has been greatly enhanced by using these plots.

To adjust for the difference in the area ratios obtained here and the ones used in the reference solution (NASA's One Dimensional Equilibrium), a program is used to find the specific impulse according to the area ratio measured before and after the run.

A new means of determining the proper coefficients as input to the efficiency program is now being used. A weighted linear regression program supplied by the Computer Science Center is a very precise way to perform this curve-fitting operation for the theoretical C* and Isp equations.

Efficiencies for runs 28 to 32 were computed. The results were plotted and are now being analyzed.

A new remote batch processing computer input terminal has been installed at the JPC. The use of this terminal for many small, intermediate programs such as the linear calibration program and area ratio as a function of specific impulse program can be run and results obtained immediately, thus speeding up the entire data reduction process.

The review and up dating of the computing methods are now complete so that in the future, all results can be reduced and plotted in final form within a week after a rocket firing.

Task 2. Heat Transfer

Various experimental techniques used to measure heat transfer rates in rocket combustion chambers and nozzles have been reviewed. In addition to the calorimeter method of performing these measurements which was outlined in an earlier report, the following additional methods have been reviewed.

1) Segmented engine
2) Ablative thermocouple
3) Passive temperature indicator

The segmented engine method basically involves construction of the engine in short, thermally isolated, axial segments, with each segment being cooled by flowing a coolant, such as water, in a radial cavity in the segment. Much more study of this method is planned. At this time the segmented engine method appears the least attractive, due to sealing problems at 4000 psi, cost, and material problems.

In the ablative thermocouple method, special thermocouples constructed with the same phenolic material as the engine liners are installed in the wall of engine at axial locations of interest. The thermocouple ablates along with the ablative material of the engine and yields a continuous wall temperature. The heat fluxes are then predicted using an approximate method outlined by Howard (1). Particular problems with this method are dependability of continuous thermocouple output as the ablation occurs, holding the thermocouple in place under pressure, and sealing them once they are held in place.

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(1) Howard, F. G.: Single-Thermocouple Method for Determining Heat Flux to a Thermally Thick Wall. NASA TN D-4737, Sept. 1968.

Passive temperature indicators replace thermocouples with encapsulated mixtures of powdered refractory metals and carbon (2). They are embedded in the wall of the ablative material and analysis of heat transfer is performed from the temperature measurements in the same general way as for the ablative thermocouple. The advantage of this method lies in the ability of the indicators to measure higher temperature than is possible with thermocouples. The primary disadvantages in this method are the trial and error experimentation needed to match the indicator composition with the application and the non-continuous temperatures obtained (the maximum temperature is estimated from post run radiograph and metallurgical analysis of indicators).

Mr. Warren Brecheisen is undertaking the heat transfer phase of the high pressure rocket program as a Ph. D. thesis. Mr. Brecheisen received his B. S. in Mechanical Engineering at Kansas State University in 1967. As an undergraduate, he participated in the Student Co-op program with NASA at the Marshall Space Flight Center. He received his M. S. in Mechanical Engineering in 1969 at Kansas State University.

Task 3. Combustion Stability Investigation

During September and October no experimental runs were conducted for stability analysis. This delay was due to excessive electronic noise levels on the four channel Honeywell LAR 7400 analog tape recorder, the primary high frequency data recording instrument for the combustion stability analysis. After extensive trouble shooting efforts by JPC personnel and many consultations with manufacturer to isolate the source of the noise, it was concluded that the recorder heads needed to be relapped by the manufacturer. This was done and the heads were re-installed. This reduced the noise to acceptable levels. In an attempt to reduce program delays an extensive search was made for a temporary replacement machine; but with no success.

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(2) Ellison, J. R. and Binder, H. I.: Evaluation of Rocket Nozzle Passive Temperature Indicators. Tech. Rep. AFRPL-TR-69-69, U. S. Air Force, April, 1969.

During the first week in November a successful experimental run for stability analysis was achieved. Therefore a brief discussion of the preliminary results will be included in this report. This run was made using a new 0.889" diameter ablative throat. Low frequency instability, occurred throughout the run. This run was made using the large droplet injector with tube dimensions of:

oxidizer orifice dia. = 0.118 inch

fuel orifice dia. = 0.106 inch

For the observed steady state pressures:

Chamber pressure P_c = 3800 psi

Ox Manifold pressure = 4200 psi

Fuel Manifold pressure = 4150 psi

For this run the pulse gun was assembled with 10 RDX wafers (56.4 grains) recessed 1 11/16 inch. A 1/2 inch ablative plug and zinc chromate putty were used to fill the space between the combustion gases and the explosive charge and thus provide additional thermal protection. The pulse gun was not pressurized and no retaining pin was used for the ablative plug. The pulse gun fired successfully at the programmed time. The peak-to-peak pressure fluctuation recorded was 5270 psi and the pressure fluctuation damped completely in 0.003 second, the analog tape playback is shown in Fig. 3. The photocon pressure transducer trace immediately resumed the oscillatory motion characteristic of chugging.

This successful pulse gun firing shows promise for the design modification reported previously in the six months progress report.

Capt. Thomas L. Larsen has began work on his Ph. D. Thesis in the Combustion Stability area of this program. Capt. Larsen received a Master of Science Degree in Aerospace-Mechanical Engineering with a Gas Dynamics-Propulsion Major at the Air Force Institute of Technology Resident School of Engineering, Wright-Patterson AFB, Ohio in June 1965. Since then he has been assigned as a project engineer at Air Force Flight Dynamics Laboratory's 50 Megawatt Electric Arc-Heated Hypersonic

Wind Tunnel. Capt. Larsen personally directed the final construction and checkout of the 50 Megawatt Facility. He was a member of the 5-man team who increased the operating power level of vortex stabilized high voltage electric arc heaters by a factor of 10 in a period of 2 years. For the past year he served as project engineer for the development and checkout of the Re-Entry Nose Tip (RENT) test leg in the 50 Megawatt Facility. The RENT test leg utilizes the same arc heater for high shear-high transfer testing of advance missile nose tips.

During this report period Capt. Larsen initiated a review of available literature concerning this task and has started a program of familiarization with Jet Propulsion Center Combustion Research Facility and, in particular, with the hardware associated with the high pressure rocket program. This effort has included participation in engine assembly and checkout, sequencing of automatic controls, and propellant transfer operations.

During October this researcher participated in an experimental run which was conducted for analysis of the performance of the solid copper nozzle coated with zirconium oxide. This experience provided knowledge of test operations and facility control, as well as knowledge of the remainder of the data acquisition and recording system.

III. Future Plans

Current pulsed firings of the high pressure rocket engine show that three milestones of the Combustion Instability Investigation have been met.

1. The pulse gun design modification for high pressure application appears to be successful. Further minor modifications should decrease the present difficulty in post-fire disassembly.
2. The pulse gun sequence triggering system is performing well.
3. The pressure sensing instrumentation, consisting of a photocon pressure transducer whose output is recorded on the analog recorder and played back onto the visicorder oscillograph, is properly functioning and recording the

sharp pressure pulse produced by the gun.

These developments allow the resumption of the original plan to determine the combustion stability bounds at 4000 psi chamber pressure. It is planned that three pulsed firings will be made for each of three mixture ratios (O/F = 1.8, 2.0, 2.2), two injector configurations (large and small droplet size),, and two chamber characteristic lengths (L* = 50, 100). Thirty-six firings are anticipated as necessary to accomplish this task.

Data from this series of runs will be used to map the stability bounds as functions of burning rate parameter, and viscous dissipation parameter.

Work in the area of heat transfer determination will be concerned primarily with the examination of the adaptability of the single thermocouple method to an ablative material. The mechanisms of ablation and the equations used to describe these mechanisms will be studied. A scheme will be sought which, when introduced into the single thermocouple method, will account for the variable thicknesses associated with ablation and the change from unexposed material to char material.

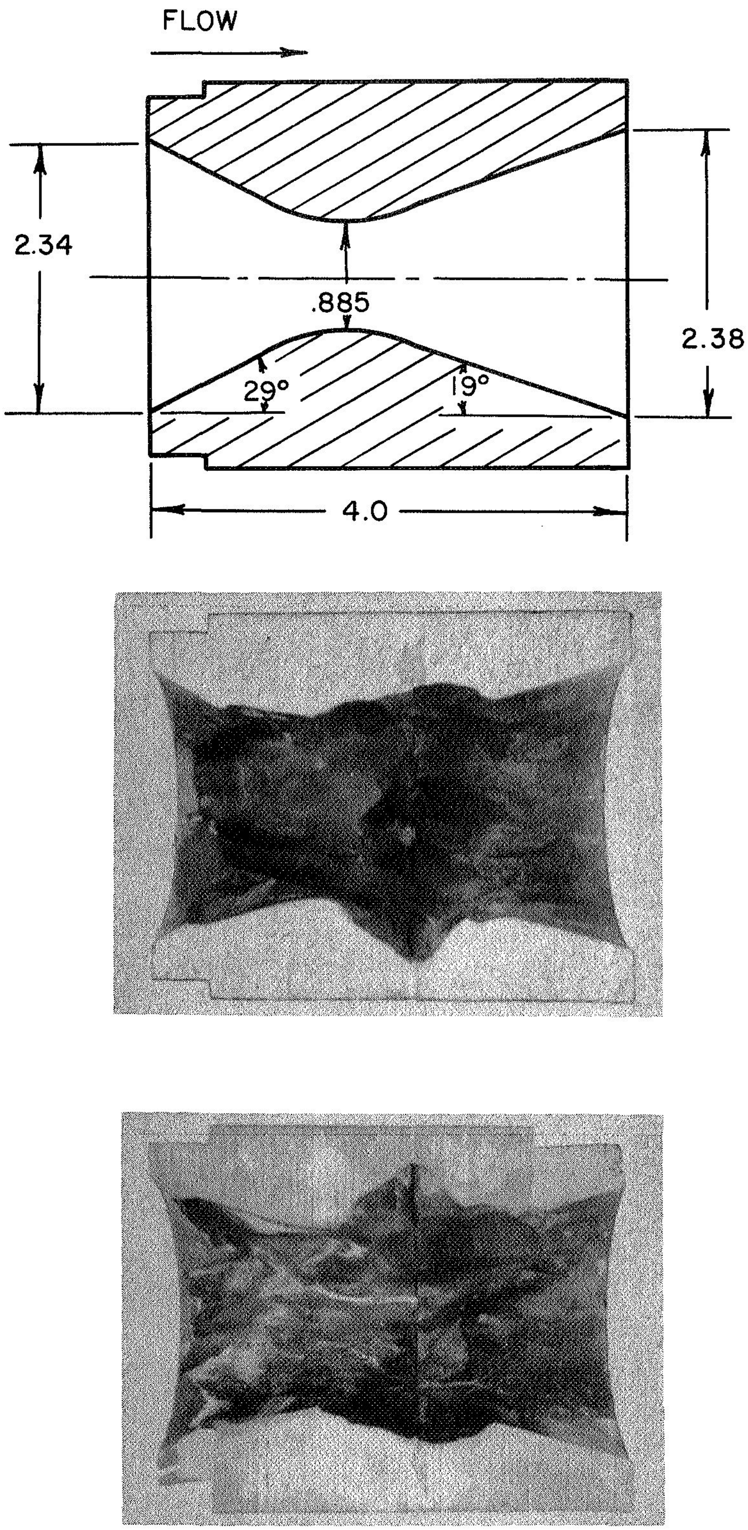

FIGURE 1 CROSS-SECTION OF ZIRCONIUM OXIDE COATED COPPER THROAT

NOTE: COPPER BASE WAS ETCHED TO SHOW GRAIN BOUNDRIES

MAGNIFICATION = 150X

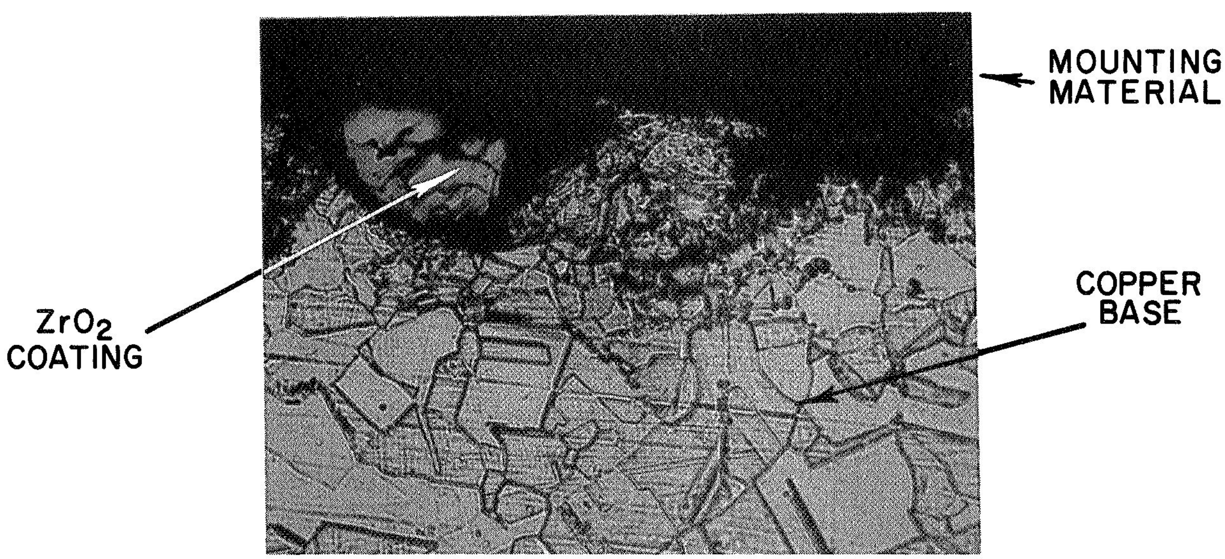

FIGURE 2
MICROGRAPHS OF POST FIRE NOZZLE SECTION

FIGURE 3 RUN 41 PHOTOCON PULSE TRACE